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DETAILED MISSION PLANNING CONSIDERATIONS AND CONSTRAINTS

By M. P. FRANK III
Mission Analysis Branch

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MISSION PLANNING AND ANALYSIS DIVISION
MANNED SPACECRAFT CENTER
HOUSTON, TEXAS

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PROJECT APOLLO

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DETAILED MISSION PLANNING CONSIDERATIONS

AND CONSTRAINTS

By M. P. Frank, III

PREFACE

This report was originally presented as a speech at the Apollo Lunar Landing Mission Symposium at the Manned Spacecraft Center in June 1966. The speech was presented in parts at two sessions of the symposium, and, hence, this report is in two parts.

CONTENTS

FIRST SESSION: LAUNCH THROUGH EARTH ORBIT INSERTION

INTRODUCTION

The purpose of this paper is to present the various constraints that affect the lunar mission planning, primarily in the form of trajectory shaping and the limitation to launch opportunities.

Trajectory geometry constraints and spacecraft performance capability combine to limit the accessible area on the moon. Accessible area limitations combine with operational constraints to limit launch opportunities to certain specifically defined periods. To understand the mission planning considerations and to appreciate the effects of the various constraints, one must closely examine the trajectory characteristics. In this paper an attempt is being made to explain the interrelation of constraints and trajectory shaping. Unfortunately, the explanation becomes quite detailed for some phases of the mission, but this is considered necessary in order to obtain an understanding of the interrelation.

We will begin with the launch and work through the mission, phase by phase. This paper will not describe every trajectory shaping consideration. It will only hit the highlights and discuss the more significant considerations - those that have a major effect on mission planning and the determination of launch windows.

LAUNCH PHASE

The mission planning considerations of the launch phase of the lunar mission are primarily related to launch windows, booster performance, and contingency planning. Launch windows are defined for two different time intervals. A daily window has a duration of a few hours during a given 24-hour period. A monthly window consists of a few days during a given month or lunar cycle. The daily window is continuous from opening to closing, but a monthly window may have gaps. For example, a monthly launch window may cover a 7-day period, but a daily window may not exist for some of the intermediate days. A description of the factors that define the launch windows will be presented later in this session. For now, only the effects on the launch phase of providing a window will be considered.

It is obvious that for operational flexibility it is highly desirable to have both daily and monthly launch windows as large as possible. A daily window allows delays or holds in the countdown. The mission would not have to be rescheduled for another day if the window was larger than the cumulative delay or hold times. A monthly launch window allows the mission to be quickly rescheduled. If the daily window was missed, the mission would not necessarily be delayed for a month.

Although the duration and frequency of holds is strongly dependent on the actual vehicle, some estimation of the expected holds can be obtained from historical data. For programs employing the Atlas, Titan, and Saturn launch vehicles, the data shows that only in rare cases is a hold of greater than 2 hours followed by a successful launch. This indicates that a 2-hour window should be adequate. If the hold were to exceed 2 hours, the mission would probably have to be scrubbed anyway.

For the lunar missions, daily launch windows require changes in launch azimuth: the larger the daily window, the larger the required azimuth change. The mechanism by which variable azimuth provides launch windows will be described in the translunar injection phase. For now, let's assume it is required. The limitations to the launch azimuths that can be used are based on considerations of range safety, booster performance, and insertion tracking requirements.

1. Range safety.- In the early days of lunar mission planning, the range safety limits were defined as 72° and 108°; however, there is some indication now that these could be increased if necessary. The primary concern of these range safety limits is to keep the space vehicle on the range following any aborts during launch.

2. Booster performance.- A 90° launch azimuth takes maximum advantage of the earth's rotation in achieving orbital velocity. As the azimuth is shifted away from 90°, the booster performance

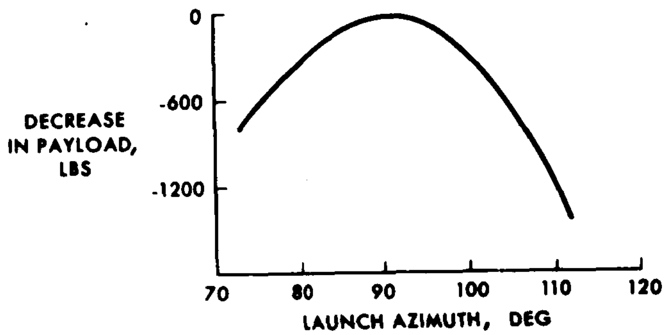

Figure 1.- Effect of launch azimuth on payload capability.

requirements are increased, or its payload capability is decreased, as indicated in figure 1. The Saturn vehicle is capable of providing lunar mission payloads for launch azimuths beyond the 72° to 108° range; however, its flight performance fuel reserves are drastically reduced. For this reason, the mission will be constrained to these launch azimuth limits.

3. Insertion tracking.- There is an operational requirement to track the space vehicle from orbit insertion to at least 3 minutes following insertion in order to make a GO/NO-GO decision. Since the Apollo space vehicle is inserted into orbit some 1400 n. mi. downrange, a ship is required to provide this tracking. The ship speed is relatively slow compared with the launch azimuth change during a launch window, and it cannot keep up with the changing ground track. Thus, the tracking coverage downrange afforded by one ship limits the range of usable launch azimuths to 26°, as shown in figure 2 on the following page. A 26° range of usable launch azimuths anywhere between the values of 72° to 108° provides at least a 2.5-hour daily window. The choice of where the 26° range is located within the maximum bounds is left up to the mission planner, and is based on such things as maximizing spacecraft fuel reserve, MSFN tracking coverage, launch window duration, and providing a daylight launch.

Another constraint on the launch phase of mission planning is the monthly launch window. A monthly launch window allows the mission to be rescheduled as soon as possible in case it is scrubbed for any reason on a given day or in case a hold extends beyond the daily window. It also allows some flexibility in the initial planning of the launch day.

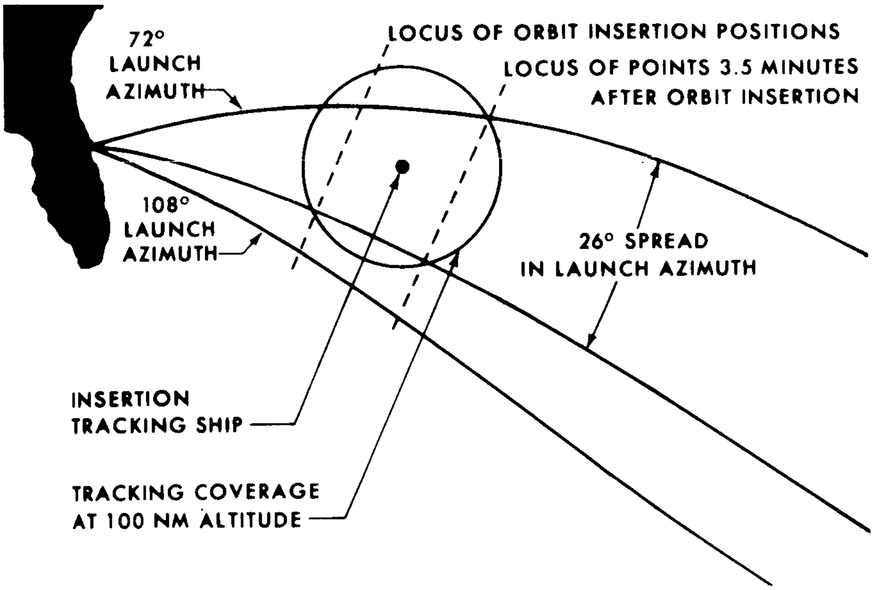

Figure 2.- Orbit insertion.

How monthly launch windows are obtained and their limitations other than vehicle systems is the subject of the rest of this session. After describing the various constraining considerations during the rest of this talk we will come back to this particular subject. For now, we will discuss only the effects of vehicle recycle characteristics on the requirements for and limitations to monthly launch window selection.

The minimum turnaround time, or the space vehicle recycle time, (fig. 3), is a major factor in defining the minimum acceptable duration of the monthly launch window. Studies for NASA Headquarters by Bellcomm have provided some significant data that have been used to develop the minimum launch window philosophy.

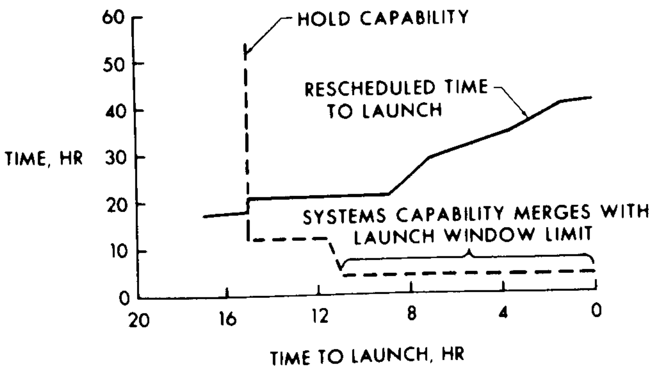

Figure 3.- Launch ability by hold or recycle.

If the mission is scrubbed after the countdown has reached T - 6 hours, the minimum time to recycle is in excess of 30 hours, and is as long as 40 hours at T = 0. Thus, a minimum window required to guarantee a recycle capability is

3 days, which does not allow any additional time for repairs or replacing components. If this activity could not be done in parallel with the recycling, 3 days would not be sufficient. Therefore, a window of only 3-days duration is not desirable, but is a minimum. In order to allow time for repairs and still make the monthly launch window, it should be as long as possible. The Bellcomm studies indicate that the probability of a successful launch is between 85% and 90% if a 3-day window is available, and that this increases to about 95% if the launch window is of 5-day duration. Based on this data, the lunar landing mission is being planned only for those periods when at least a 3-day window exists, and every effort is being made to provide 5-day windows.

One final consideration in the launch phase is the desirability of a daylight launch. There are three reasons which make a daylight launch highly desirable, all of which are concerned with contingencies.

1. In the event of an abort off the pad, the recovery of the crew in the Merritt Island area would be complicated if it had to be performed under conditions of darkness.

2. Aborts later in the launch require attitude maneuvering of the SC, for which it is desirable to have a sunlit horizon as a backup for attitude reference.

3. Finally, it is desirable to have photographic coverage of the boost phase for postflight analysis if a catastrophic failure occurs.

Because of these three considerations, every effort will be made to provide a daylight launch, although the mission will not be constrained to daylight launches only.

EARTH PARKING ORBIT PHASE

Earth parking orbits are required in order to provide launch windows of reasonable duration. Direct lunar injections are possible; however, the launch windows are unacceptably small.

The only major consideration in the earth parking orbit phase is the duration or the number of earth orbits. The parking orbit duration is constrained by space vehicle systems considerations. The maximum duration is 4.5 hours from orbit insertion to the beginning of injection and is limited by the launch vehicle capability to provide attitude control and by the battery lifetime. This time allows up to three parking orbits prior to the second S-IVB burn. There are other considerations in limiting the parking orbit duration, although they are not hard constraints. The S-IVB propellant boil-off and inertial platform drift make it desirable to inject as soon as possible.

The minimum duration of the earth parking orbit phase is limited by the time required to perform system checks and realign the spacecraft platform. Analysis of the crew's schedule indicates that this will require at least 1.5 hours.

There is also a minimum network coverage requirement which states that two tracking stations and a command station must be passed before the GO decision for the second S-IVB burn; however, this is always accomplished in the first orbit. Thus, there is a 3-hour period from 1.5 to 4.5 hours after orbit insertion in which the translunar injection can occur. This means that the injection must occur on the second or third orbit. Figure 4 illustrates the ground tracks for three earth orbits for a typical launch azimuth. The solid line indicates that part of the orbit on which a translunar injection could occur.

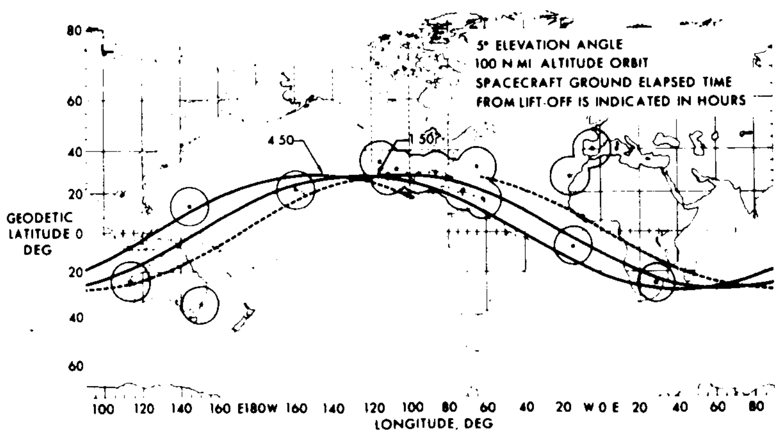

Figure 4.- Earth parking orbit ground track for three orbits for a 90 degree launch azimuth.

TRANSLUNAR INJECTION

The translunar injection point is rigidly constrained by performance considerations; the geometry of the moon's orbit, the energy requirements of the earth-to-moon transfer trajectory, and the necessity of efficiently burning the S-IVB propellant all combine to place very tight restrictions on the location of the injection maneuver. An attempt will be made in this section to show how these considerations are interrelated, and how this injection position is determined. It is somewhat involved and will take considerable explanation.

The translunar trajectory is essentially a highly eccentric elliptical earth orbit. The translunar injection maneuver is an orbital transfer which takes the spacecraft from the circular orbit to the elliptical orbit. In order to arrive in the vicinity of the moon, the spacecraft is aimed at a position where the moon will be at the time of arrival, as illustrated in figure 5. In order to accomplish this rendezvous with a minimum expenditure of propellant, the injection must occur very close to the extension of the earth-moon line at the time of arrival. (The negative of the unit vector of the moon's position is called the moon's antipode.) Something closely akin to a Hohmann transfer is what is being strived for.

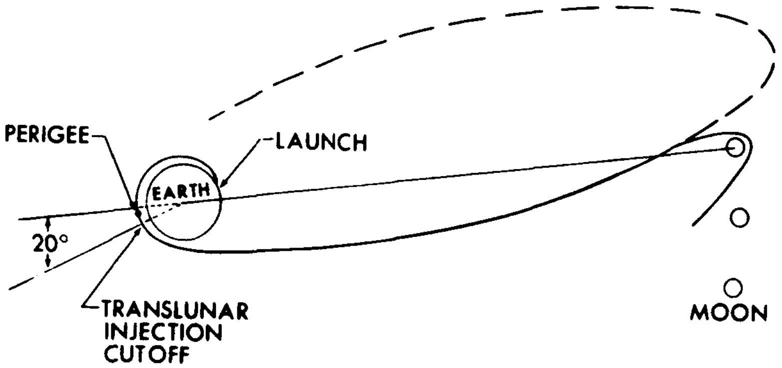

Figure 5.- Translunar injection geometry.

A minimum energy transfer would place the perigee on the antipode if the moon's mass did not perturb the trajectory. However, the moon does perturb the trajectory, as shown in figure 5, and the perigee must lead the antipode by approximately 8° to compensate for the perturbation. The apogee altitude of the osculating conic trajectory is determined by translunar flight time, which defines the trajectory energy required.

To inject to the moon in the most efficient manner, an impulsive velocity would be added along the orbital velocity vector, resulting in an injection at perigee on the translunar conic trajectory. Since an impulsive addition of velocity is not possible, a finite burn time is required, and the actual injection position is on the order of 20° ahead on the antipode. The thrust is directed approximately along the velocity vector, and as the speed increases above orbital speed, the altitude and flight-path angle also increase. For the Apollo configuration, by the time a sufficient energy increase is realized, the altitude increases 60 n. mi. above the orbit and a positive flight-path angle of 6° to 7° has been gained. Since the conic trajectory is very nearly parabolic (eccentricity ≈ 0.97), the true anomaly is approximately equal to twice the flight-path angle, so perigee is approximately 12° to 14°

behind the burn cutoff position. The burn arc itself is 25°, so that ignition always occurs within a few degrees of the antipode.

The preceeding discussion has shown that the injection position is very closely related to the moon's antipode. To go to the moon efficiently the spacecraft must inject near it, so that we must now consider the problem of getting to the antipode from the launch pad at the Cape.

The antipode, which is a unit vector from the center of the earth in the direction negative to the moon's position, moves as the moon travels in its orbit. The launch pad is rotating with the earth, and both of the motions must be compensated for in order to rendezvous with the antipode. It is convenient to divide the description of the antipode movements into two categories - a long period cycle and a short period cycle.

The long period cycle is due to the moon's orbital travel about the earth Figure 6 illustrates this effect. Assume that the earth is a fixed, motionless sphere and is not rotating about its axis. The moon's orbit plane cuts this sphere, as shown. As the moon revolves around the earth, its antipode would trace a great circle in this plane around the surface. Note that the direction of travel is from west to east. The orbital period is some 28 days, and thus at the end of this time the antipode would be back where it began. The rate of travel of the antipode is about 0.54° per hour. The latitude of the antipode has a time history similar to that shown in figure 6.

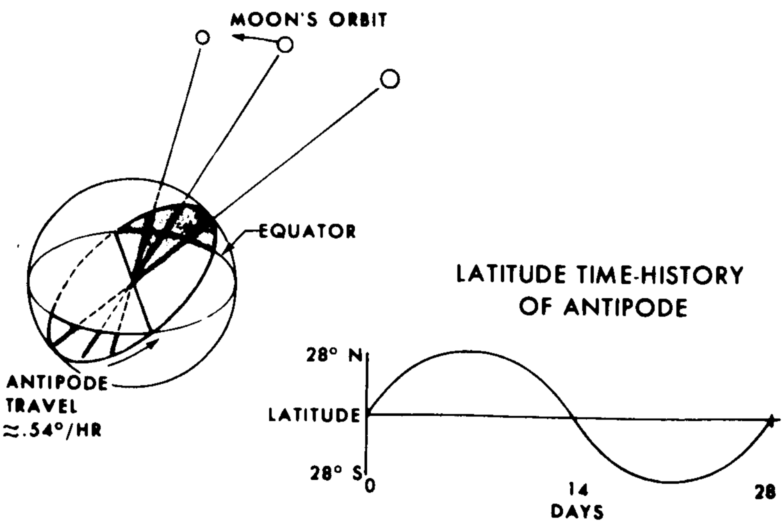

Figure 6. - Long period motion of antipode.

The short period motion of the antipode across the surface of the earth is due to the earth's rotation. To illustrate this, it is assumed that the moon is fixed at some position in its orbit, and the earth is now allowed to rotate about its polar axis. The antipode travel is illustrated in figure 7. In this case the latitude is constant, and the longitude changes from east to west by 15° per hour. The complete picture of the antipode travel across the earth's surface is obtained by combining the long period and the short period motions. The latitude varies sinusoidally with time with an amplitude of 28.5° (in 1968) and a period of 28 days. The longitude varies by a nearly linear rate of 14.5° per hour.

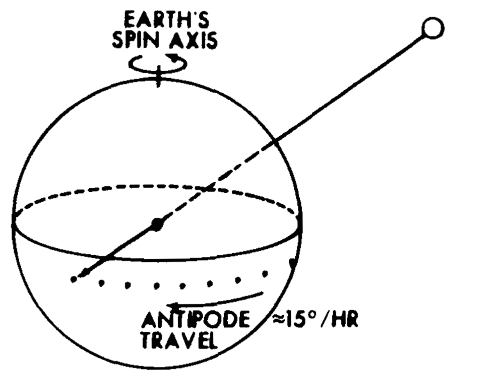

Figure 7.- Short-period motion of antipode.

The launch must occur at a certain time for each launch azimuth in order to intercept the antipode. The correct launch time is defined by the antipode's position, the time interval from launch to arrival at this position, and the antipode travel during this time interval. Figure 8 illustrates this problem. Consider an inertial sphere having a radius equal to the earth's radius. A plot of the launch pad travel as a function of time on this sphere is represented by a fixed latitude completely encircling the sphere. The launch pad completes one revolution per day. The trace of the antipode is given by the intersection of the moon orbit plane and the sphere. The antipode completes a revolution every 28 days. The launch for any given

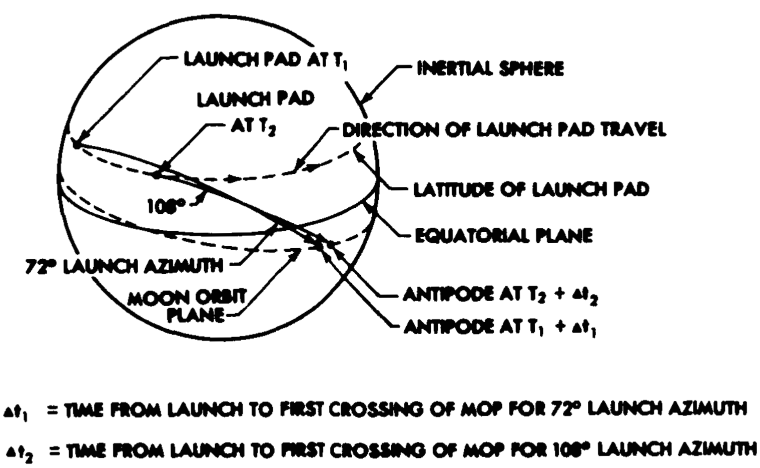

Figure 8.- Antipode rendezvous geometry.

azimuth must be timed so that the inertial plane of the resulting orbit contains the antipode at the time the space vehicle crosses the moon orbit plane. Later launch times require greater launch azimuths. If additional parking orbits are required, the launch must occur later to account for the additional antipode travel. For each 360° travel of the launch pad, there are two launch times for each azimuth which allow interception of the antipode. This is better illustrated in figure 9, which shows the same situation in earth-fixed coordinates.

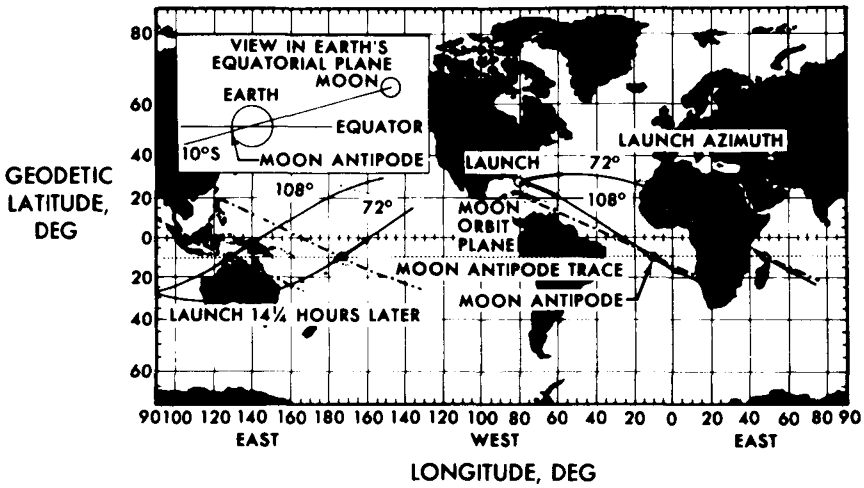

Figure 9.- Translunar injection characteristics.

In figure 9, the launch pad is now fixed, and the antipode travels rapidly over the surface of the earth. The antipode position is shown at four different times during the day, corresponding to the positions at intercept for 72° and 108° launch azimuths. The launch must be timed so that the vehicle intercepts the moving antipode. The time required for the antipode to travel from the interception of the 72° launch azimuth trajectory to the interception of the 108° trajectory defines the launch window duration. The intersection of the MOP is drawn in for each position. This figure shows how two different launch times for one azimuth can provide intercept with the antipode. One provides injections going south over the Atlantic Ocean and the other provides injections going north over the Pacific Ocean. For the day illustrated in the figure, the Atlantic injection gives a trajectory that is nearly in the moon's orbit plane, and the Pacific injection results in a trajectory that is highly inclined to the moon's orbit plane. Half a lunar cycle later, the Pacific injection would be highly inclined. The magnitude of this relative inclination depends on the lunar declination and is a maximum when the moon is near the equator. When the moon is near maximum declination, both windows provide trajectories with low relative inclinations.

It is of special significance that the Pacific injection always results in a trajectory above the moon's orbit plane, regardless of the moon's declination or whether it is ascending or descending. This effects the relative location of available landing areas on the moon from these two injection windows. This effect will be described later.

It can be seen that when the launch azimuth bounds are defined, the proper launch time can be found, allowing for the number of parking orbits to be employed prior to injection.

If for some reason the injection opportunity were missed, injection could be attempted one orbit later when the space vehicle again approached the antipode. However, since the antipode is traveling in a plane that is not necessarily the same as the vehicle's orbit, a plane change would be required. This is illustrated in figure 10. It can be seen that the antipode has traveled out of the parking orbit plane when the vehicle returns to the position of injection. The magnitude of the out-of-plane travel is dependent on the relative inclination between the parking orbit and the moon orbit planes. The maximum value is about 0.6°. This second injection would require a greater propellant expenditure by the S-IVB because of the plane change involved. If two injection opportunities are to be provided, the launch would be timed so that both would require a plane change, because this minimizes the propellant required. The launch would occur a little bit later so that the first time the vehicle crosses the moon orbit plane, the antipode has not reached the parking orbit plane. The second time the vehicle crosses the moon orbit plane, the antipode has passed through the parking orbit plane. If three injection opportunities are to be provided, the launch would be timed so that the antipode was in the parking orbit plane for the second one.

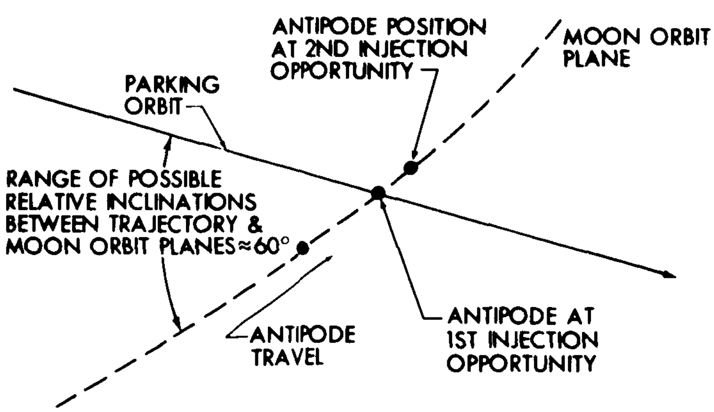

Figure 10.- Effect of missing 1st injection opportunity.

Figure 11 illustrates the effects of different targeting methods on the characteristic velocity required to provide additional injection opportunities. Three cases are shown. The first shows additional ΔV required when the launch is timed for the first injection to be coplanar. The second and third opportunities have large additional ΔV requirements.

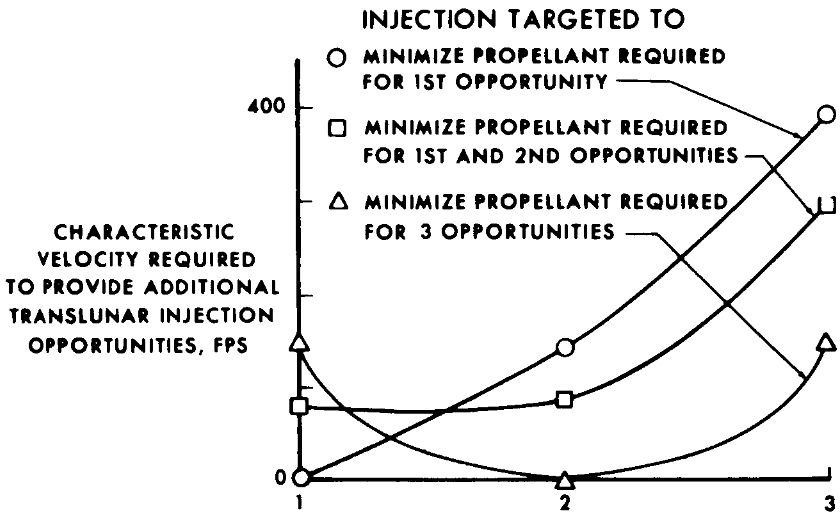

Figure 11.- Injection opportunity.

In the second case, the launch is timed to split the delta azimuth between the first and second injection opportunities. This would be used for two injection opportunities.

In the third case, launch is timed so that second injection opportunity is coplanar. This method would be used to provide three injection opportunities.

The penalties shown are only illustrative; the actual values strongly depend on relative inclination between the two planes.

Because injection is limited to the second or third orbit, only two injection opportunities are planned for 504 mission, and the second targeting technique is being used.

The combination of launch azimuth limits, parking orbit duration constraints, and the geometry of the moon's orbit confine the location of the injection positions to two geographical areas. These areas are generally centered over the south Atlantic Ocean and the Pacific Ocean, and for this reason are distinguished by these names. The bounds, as shown in figure 12, are defined by the first orbit for a 72° launch azimuth, the third orbit for a 108° launch azimuth, and the extremes of lunar declination. The areas shown in figure 12 contain all of the possible injection positions.

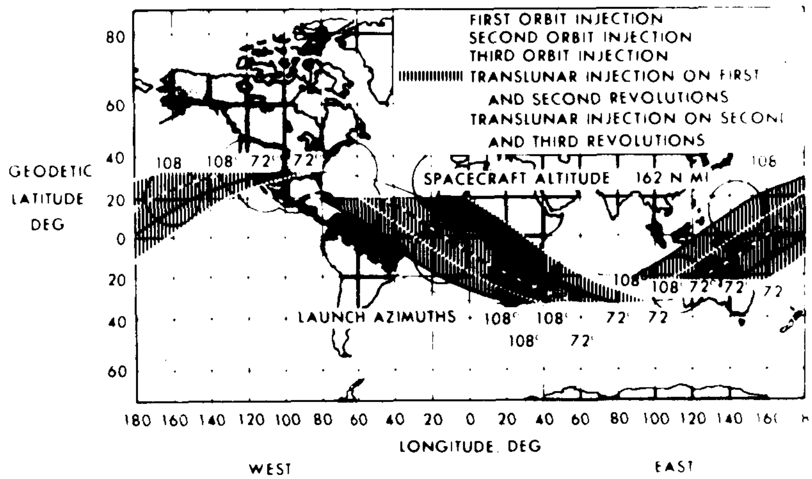

Figure 12.- Envelope of free return translunar injection cutoff positions for 1968 and 1969.

TRANSLUNAR COAST PHASE

For the translunar coast phase, the description will be confined to the effects of the trajectory inclination relative to the moon's orbit plane, the effects of the free-return flight plan and its relationship to the lunar orbit insertion maneuver, and finally a discussion of some alternatives to the free-return flight plan.

The first point to be made in describing the translunar coast trajectory is in regard to the relative location of the trajectory to the moon's orbit plane. It was stated earlier that Pacific injections always result in translunar trajectories above the lunar orbit plane and that Atlantic injections always result in translunar trajectories below it, as shown in figure 13 on the following page. The amount of out-of-planeness is a function of the moon's declination and whether or not it is ascending or descending in its orbit. These parameters influence the magnitude of the effects; however, they do not change the general conclusions.

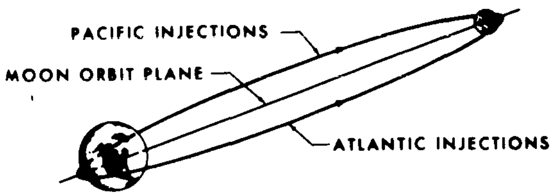

Figure 13.- Effect of injection window on translunar trajectory geometry.

Following a Pacific injection, the spacecraft approaches the moon from above the moon orbit plane. This forces the trajectory below this plane on the far side of the moon, where the lunar orbit insertion maneuver takes place. The resulting lunar orbit then is constrained to be approximately as illustrated in figure 14. A plane change during the orbit insertion can modify the resultant orientation somewhat, but the basic conclusion can still be drawn that to land at northern latitudes on the front side of the moon, a Pacific injection will result in lower propellant costs. Conversely, Atlantic injections favor the southern latitudes. This will be clearly demonstrated later when the accessible lunar areas are defined.

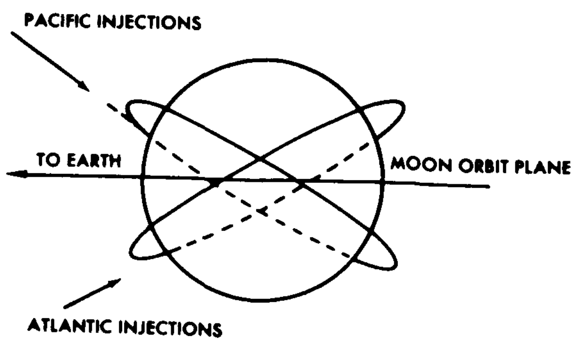

Figure 14.- Effect of injection window on lunar approach conditions.

FREE RETURN

One of the most constraining requirements of the lunar landing mission is the requirement for a free-return trajectory. It severely limits the area on the moon to which Apollo missions can be conducted. Although it is costly in terms of spacecraft performance requirements, the inherent safety feature of a free return makes it a highly desirable method of getting to the moon.

A circumlunar free-return trajectory, by definition, is one which circumnavigates the moon and returns to earth, as shown in figure 15. The perigee altitude of the return trajectory is of such a magnitude that by using negative lift the reentering vehicle can be prevented from skipping out of the atmosphere, and the aerodynamic deceleration can be kept below 10 g. Thus, with a complete propulsion system failure following the translunar injection, the spacecraft would return safely to earth.

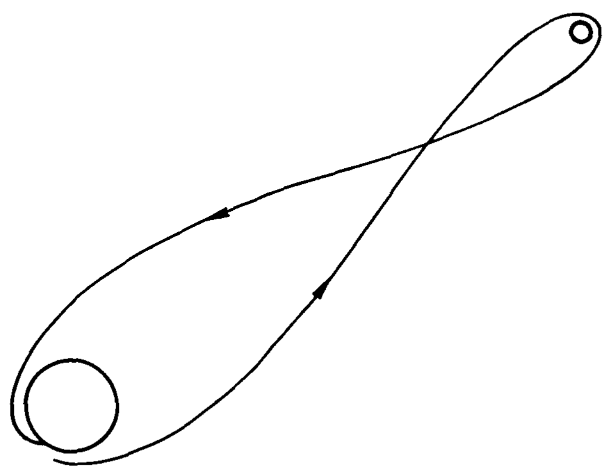

Figure 15.- Free return trajectory

The range of return-perigee altitudes that provide this feature is called the reentry corridor and is primarily a function of the lift-to-drag ratio of the reentry vehicle. For the Apollo vehicle this corridor is approximately ± 12 n. mi. centered around a 25-n. mi. altitude. The injection velocity accuracy required to achieve a free-return trajectory is less than a 0.1 fps. Obviously, this is well beyond the capability of any guidance system when the total ΔV involved is on the order of 10 000 fps. However, it is still valid to plan for a free-return trajectory, because this procedure at least will minimize the

ΔV requirements to return to earth should a SPS failure occur. In this situation, there is a good probability that the RCS can provide the necessary velocity corrections to overcome the injection errors.

The free-return trajectory severely limits the accessible area on the moon because of the very small variation in allowable lunar approach conditions and because the energy of the lunar approach trajectory is relatively high. The high approach energy causes the orbit insertion ΔV to be relatively high. However, the main limitations to accessible area are a result of the small range in flight times from earth to moon. Figure 16 illustrates the effect of flight time on the location of perilune. All free-return trajectories have translunar transit times between 60 and 80 hours, and it can be seen in figure 16 that perilune is limited to a region within about 10° of the negative of the earth-moon line or approximately 180° longitude. For non-free-return trajectories, the transit time can be anything from 50 to 110 hours.

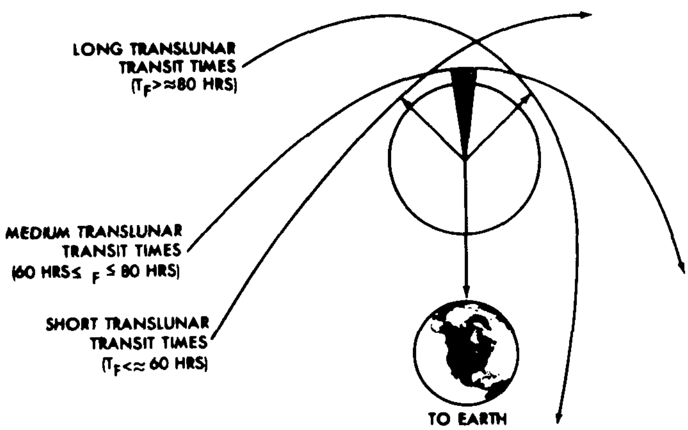

Figure 16.- Perilune position for different translunar transit time. (Atlitude of perilune $\approx$ 80 n. mi.)

Perilune could be adjusted from 140° W longitude to 140° E longitude merely by selecting the appropriate flight time. This narrow region of perilune position of free-return trajectories combined with a small range of approach inclinations is what limits the accessible area.

The relative inclination between the free-return trajectory and the moon orbit plane is less than 11°. Any trajectory with a greater inclination than this simply does not return to the entry corridor at earth regardless of the perilune position. The range of free-return trajectory conditions near the moon is illustrated in figure 17. Note the relatively small cone formed by the locus of perilune positions.

The braking maneuver to decelerate the spacecraft from the hyperbolic approach trajectory to a lunar orbit is performed at or near perilune. For illustrative purposes, it will be assumed that deceleration occurs at perilune. In order to land at a site that is not contained by the approach trajectory plane, a plane change must be made. It is generally more efficient to combine this plane change with the deceleration at orbit insertion. When the landing site is near the node, however, an excessively large plane change is required to cause the trajectory to pass over the site. This is illustrated in figure 18. Since the approach trajectories have low inclinations and orbit insertion occurs near the 180° longitude, it can be seen that to cause the lunar orbit to pass over sites at high latitudes in the region near 0° longitude, large plane changes would be required. The propellant capacity of the spacecraft limits the magnitude of the plane change that can be made.

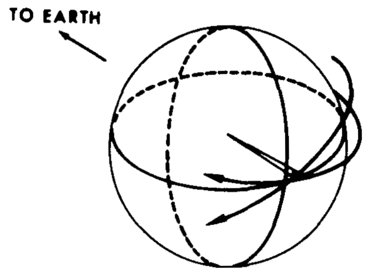

Figure 17.- Lunar approach geometry of free-return trajectories.

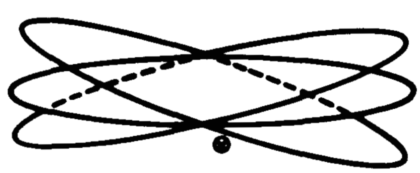

Figure 18.- Effectiveness of plane change at lunar orbit insertion on increasing accessible landing sites.

As was noted in figure 17, there is a locus of perilune positions; it is not that there is not one focal point through which all of the trajectories must pass; there is an area. This tends to relieve the limitations slightly, but the fact remains that a plane change at deboost is relatively ineffective in achieving higher latitudes near the 0° longitude. Note also that as the landing site is moved away from the 0° longitude, the plane change requirements become much less.

If the orbit insertion were not made at perilune, the magnitude of the plane change could be reduced in many cases, as illustrated in figure 19 on the following page. In this figure, two lunar orbits resulting from orbit insertion at two different positions along the approach hyperbola are illustrated. Both pass over the landing site, and both could be acceptable.

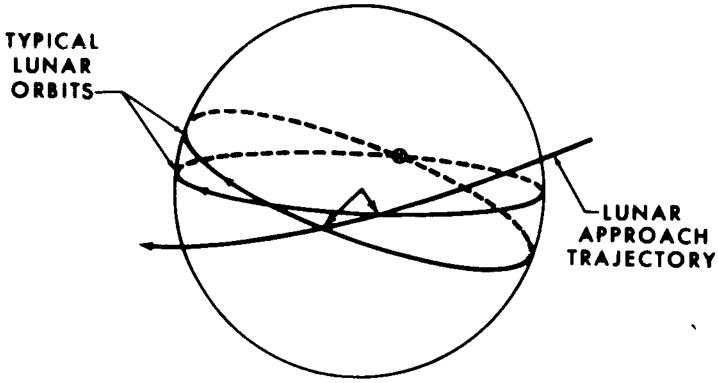

Figure 19.- Relative trajectory geometry of lunar orbit insertion maneuver.

If the insertion was performed at perilune, a much larger plane change would be required; so it appears that if the insertion was made prior to perilune, the ΔV required would be much less. However, there is an additional penalty associated with this pre-perilune braking due to the fact that a flight-path angle change must also be made. Figure 20 shows the in-plane geometry. If the deboost is performed at any position other than perilune, the velocity vector must not only be reduced, but its direction must also be changed if we are to achieve a circular orbit. A flight-path angle change is just as expensive as an azimuth change. It is much more efficient to make a small plane change and a small azimuth change than it is to make a large azimuth change. This trade-off is made in the mission trajectory design to obtain the optimum combination.

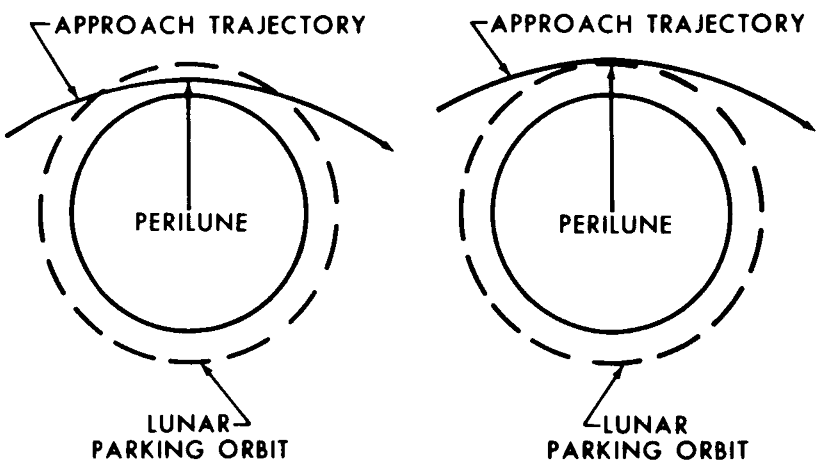

Figure 20.- Geometry of lunar orbit insertion.

Another feature of this non-perilune deboost is that the resultant orbit altitude is above the perilune altitude. The perilune must be reduced a certain amount in order to obtain the desired orbit altitude as illustrated in figure 20. The exact amount of reduction depends on the true anomaly of the deboost maneuver, but in no case is a perilune altitude of less than 40 n. mi. employed.

Since the free-return flight plan is so constraining on the accessible lunar area, parallel investigations of other techniques are being conducted. The primary goal of these parallel investigations is to develop techniques that retain most of the safety features of the free return, but do not suffer from the performance penalties. An example of this type of mission is something termed a hybrid flight plan, illustrated in figure 21. The spacecraft is injected into a highly eccentric elliptical orbit which has the free-return characteristic; that is, a return to the entry corridor without any further maneuvers. The launch vehicle energy requirements are reduced, and a greater payload (more SPS propellant) could be carried. Some 3 to 5 hours after injection, after the SPS has been checked out, a midcourse maneuver would be performed by the spacecraft to place it on a lunar approach trajectory. This lunar approach trajectory would not be a free return, and hence would not be subject to the same limitations in trajectory geometry. Landing sites at high latitudes could be achieved with little or no plane change by approaching the moon on a highly inclined trajectory. This hybrid flight plan offers large improvements in performance over the free-return plan and still retains most of the safety features. The spacecraft does not depart from the free-return ellipse until the LM docking has been completed (providing a second propulsion system for returning to earth) and then only if the SPS checks out satisfactorily.

One of the difficulties in flight planning the hybrid mission is that the initial trajectory is not amendable to conic approximations. So much time is spent milling around out near the moon that conics or

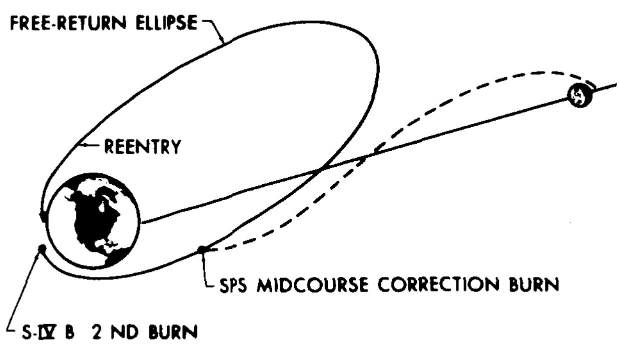

Figure 21.- Hybrid flight plan.

patched conics do not provide accurate simulations. It is extremely important that rapid calculation procedures be available because of the large number of iterations required to design a mission trajectory. If all of this must be done with precision integrating trajectory programs, the computer time becomes excessively large. Work is continuing in an effort to develop this hybrid flight plan capability. There are variations of this plan which look very promising and these are also being investigated.

A comparison of accessible area available for hybrid and free-return flight plans is given in figure 22. Only the area between 45° E and 45° W longitudes is shown, as this is the primary zone of interest.

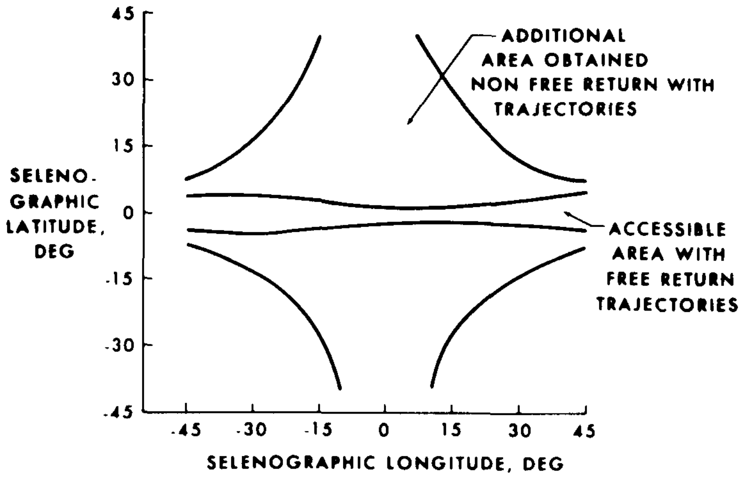

Figure 22.- Effect of non free return trajectories on 100% accessible areas.

The area available with free-return trajectories is limited near the equator. The area attainable with the hybrid mission, which is essentially the same as that for a non-free-return mission, is much larger. It includes all of the area available with the free-return, and extends to much higher latitudes at the smaller longitudes.

LUNAR ORBIT PHASE

There are only two parameters of interest in this phase. These are the orientation of the lunar orbit and the number of parking orbits required. The orientation of the plane of the lunar orbit is selected to minimize the ΔV requirements. There are three maneuvers that must be considered in this optimization. The three maneuvers are: the lunar orbit insertion, the transearth injection, and a lunar orbit plane change performed by the SM during the LM stay on the lunar surface.

The moon's relatively slow rotation rate and the low orbital inclinations result in a small out-of-plane motion of the landing site. The lunar orbit insertion maneuver is planned so that the resulting parking orbit plane contains the landing site at the nominal time of landing, as illustrated in figure 23. Position 1 represents the location of the landing site at the time of lunar orbit insertion. At the time of landing it has rotated to position 2.

During the lunar surface stay, the landing site continues to rotate out-of-plane to position 3, and in order to reduce the LM maneuvering requirements, the SM makes a plane change maneuver prior to LM launch. This maneuver is planned so that the landing site is in the new parking orbit plane at the nominal time of launch.

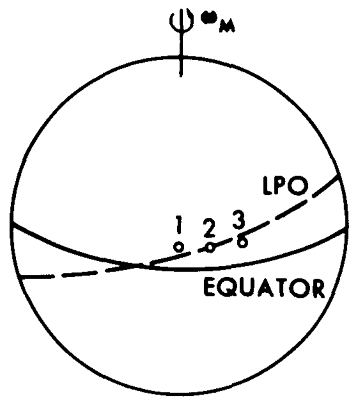

Figure 23.- Rotation of the lunar landing site relative to lunar parking orbit plane.

The transearth injection maneuver is performed from this final parking orbit orientation and, in general, a plane change is required. The SM performance requirements are minimized by selecting the best orientation of the lunar parking orbit consistant with the location of the lunar landing site and the lunar surface stay-time.

The number of parking orbits both prior to LM descent and after rendezvous are dictated by crew procedure timelines and MSFN tracking considerations. After the lunar orbit insertion, three orbits are required for the crew to activate and checkout the LM. After rendezvous, two orbits are required to prepare for transearth injection and to allow for sufficient tracking for orbit determination by the MSFN.

At this point, the sequential description of the mission planning considerations and constraints by mission phase will be interrupted. The trajectory shaping parameters have been described in sufficient detail to show the effect on launch opportunities.

TOTAL LAUNCH WINDOW CONSIDERATIONS

There are at least six major considerations that in one way or another limit the times at which the Apollo lunar landing mission can be launched. These constraining factors are due to either the characteristics of the moon's orbit about the earth, operational requirements, or spacecraft performance capability.

The constraints are listed below in the order they will be discussed.

1. Launch azimuth
2. Lighting conditions at lunar landing
3. Lunar landing-site location
4. Spacecraft performance
5. Daylight launch
6. Minimum number of launch opportunities during a month

Launch Azimuth

The mechanism of the launch azimuth effect on launch time was described earlier. It was shown that a specified launch azimuth and a specified number of earth parking orbits defined the required launch time that provided a rendezvous with the moon antipode. The fact that either of two distinct launch times would provide this rendezvous was illustrated. It was also pointed out that one of these launch times resulted in translunar injections approximately over the Atlantic Ocean, and the other resulted in injections approximately over the Pacific Ocean.

This characteristic of discrete launch azimuths defining discrete launch times can be expanded to show that a range of launch azimuths define a range of launch times. For a given range, such as 72° to 108°, the launch times for each injection window can be readily determined. These daily windows, as limited only by launch azimuth, are shown for the year 1968 in figure 24 on the following page. In this figure, the unshaded areas represent allowable launch times. The letters "P" and "A" denote the Pacific and Atlantic injection windows. Each window is opened at a 72° launch azimuth and closed at 108°.

The difference in launch time for the two windows varies throughout each month. In some periods the closing of one window is followed immediately by the opening of the other window. At other times there is as much as 14 hours between the closing of one window and the opening of the other.

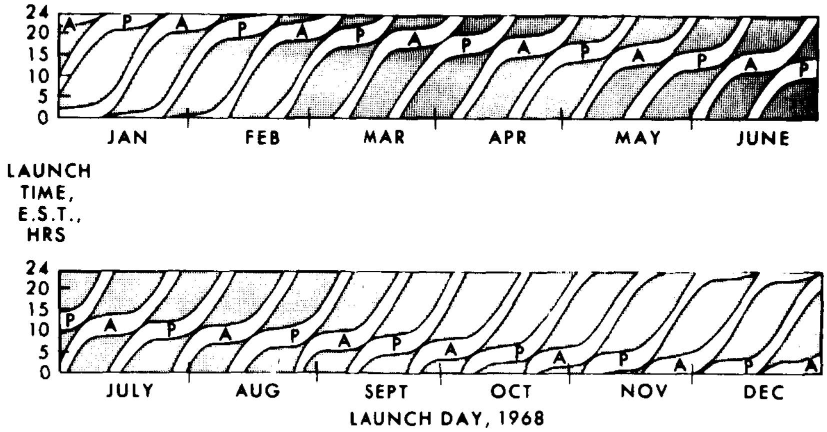

Figure 24.- Effect of launch azimuth constraints in 1968.

Note also that the time of opening of a given window is later for each successive day. The rate of change per day is quite rapid at times and at other times is almost negligible. The relatively flat period for a particular window corresponds to the part of the lunar month when translunar injection is in the moon orbit plane. This daily shift in the time of launch and differences in launch time between the two windows are important characteristics to keep in mind in the subsequent discussion.

Lighting Conditions and Lunar Landing Site Location

These two considerations are inseparable in their effects on launch windows. The effects of lighting constraints can only be evaluated in conjunction with the landing site longitudinal location, as will be seen in the following discussion.

In order to provide the LM crew with the best possible visibility conditions during landing, the sun elevation angle at the landing site during the powered descent must be between 7° and 20° above the eastern horizon. This is to allow the crew to visually evaluate the possible landing points and select a favorable one within the LM footprint.

The magnitude of the allowable range in sun elevation angle has a major effect on the determination of launch windows. Figure 25 on the following page illustrates the lighting geometry. In this figure the sunrise terminator is located approximately at 0° longitude. For this condition a lunar landing could only be accomplished in the region

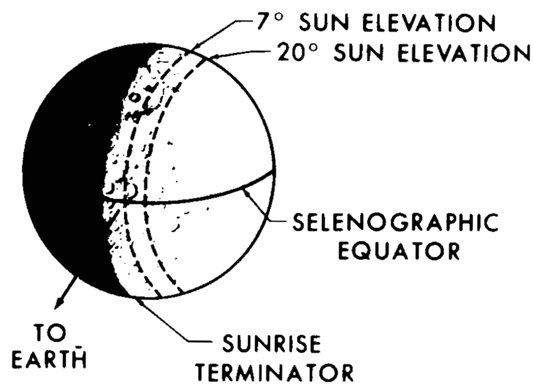

Figure 25.- Sun light conditions for lunar landing.

enclosed by the dotted lines. This region of acceptable lighting moves across the face of the moon, following the sunrise terminator from east to west at a rate of approximately 13° per day, so that the days of acceptable lighting conditions as a function of landing site longitude can be readily determined from lunar ephemeris data. The effects of latitude can be neglected in the region near the lunar equator. Figure 26 provides an example of this variation for two typical months in the first quarter of 1968. The most striking feature of this figure is the fact that for approximately 60% of the month there is no area with acceptable lighting conditions anywhere between the longitudes of 45° E to 45° W. No landing is possible from February 12 to March 3, even if the only restriction on landing site were that it must be between 45° E to 45° W.

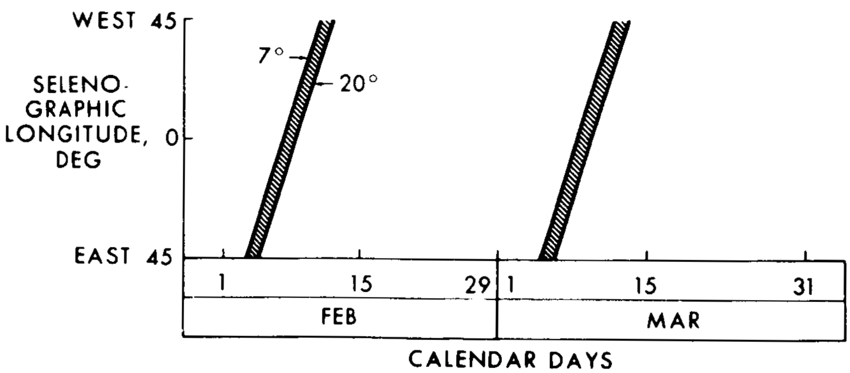

Figure 26.- Effect of lighting constraint on available landing days.

The effect of restricting landings to specific sites can also be illustrated on figure 26. For example, suppose that the only available landing site were located at 25° E; from figure 26 it can be seen that in the months of February and March, 1968, there are only two days in which a lunar landing could be accomplished - February 5 and March 5. One landing day is available during each 28-day lunar cycle. In order to provide multiple landing opportunities, several sites must be available.

The number of landing opportunities is directly related to the number of launch opportunities. For lunar landing missions employing free-return trajectories, the relationship between landing time and launch time has only a slight variation. Therefore, arrival (or landing) times can also be shown on this same figure. The landing times associated with the acceptable lighting period in February 1968 are presented in figure 27. For simplicity, only the Pacific injections are shown.

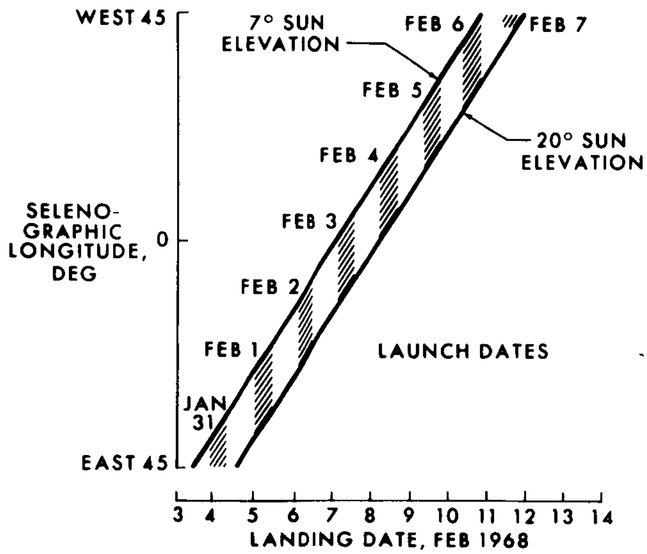

Figure 27.- Lunar landing opportunities.

Arrival times for Atlantic injections would be shifted by the difference in launch times. The launch date is noted for each band of arrival times. The variation in arrival times within a given launch window is about 10 hours and is due to the variations in launch time between launch azimuths of 72° to 108°, the possibility of injecting on the first or second injection opportunity, and the variation in the translunar transit time for different energy trajectories available in the free-return family. For a 2 1/2-hour earth launch window, this band of arrival times would be reduced to about 8 hours.

The shaded areas represent the range of landing site longitudes that have acceptable lighting for each of the launch days. Note that for each launch day, the region of available longitudes for landing is different. This region moves westward at about 13° per day, and there is virtually no overlap. No one longitude is available for more than one launch day when the requirement for a 2 1/2-hour launch window is considered. This clearly illustrates the constraining effects of lighting requirements and landing site location on the launch window.

In order to provide multiple launch opportunities during the month, several lunar landing sites must be available. One additional site is required for each additional launch opportunity. It can also be seen from this figure that to avoid duplication and gaps in the launch window the longitudinal spacing of these sites must be in increments of approximately 8° to 13°. This longitudinal spacing minimizes the probability of having 2 sites available on the same day, or what would be worse, the probability of having no site available on one day.

We can determine from figure 27, the landing site distribution required to provide monthly launch windows. For example, to guarantee that a launch opportunity exists on consecutive days the landing sites must be located between 8° and 13° apart in longitude. One site provides one launch opportunity. To provide an alternate-day launch window configuration, the sites must be located some 20° to 26° apart. To get a launch window of 5-day duration, the first and last site must be about 50° to 60° apart.

Lunar Landing Site Location and Spacecraft Performance

So far, we have only described the lighting effects on lunar landing site longitudinal location and the effect on launch dates. The latitudinal location of the landing site also has an effect on available launch dates because of its effects on service module performance requirements. The lighting may be acceptable, but if the landing site is outside the latitude bounds of spacecraft performance, then a launch opportunity still does not exist. The latitude limits which can be attained are defined by SPS propellant available as a function of selenographic longitude, lunar declination and librations, and the translunar injection window.

The accessible area for a typical day is illustrated in figure 28 on the following page. Note that the area available from the Pacific injection window is somewhat north of that available from the Atlantic window. These areas shift to the south as the moon travels to northern declinations and vice versa. In order for an acceptable launch window to exist on any given day, the landing site must be within the area defined by the latitude bounds and within the longitude region defined by the lighting bounds. The latitudinal shift in the accessible area boundaries is cyclic and has a period approximately equal to the lunar orbit period.

If a region on the lunar surface lies in the accessible area throughout the month, regardless of the daily shifting of these areas, it is said to be 100% accessible. A 100% accessible area has great significance in the selection of lunar areas to be examined for possible landing sites. If the candidate sites can be located in a region that is always accessible, then the mission planner is relieved of one very

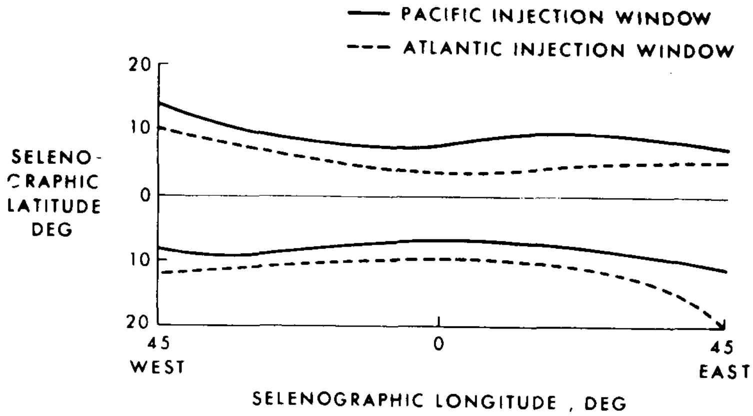

Figure 28.- Accessible lunar landing area for Feb. 18, 1968.

troublesome constraint, namely spacecraft performance. It can be guaranteed that no matter what the lunar declination or libration, when the lighting is acceptable, the landing site is attainable.

The area available every day of the month presents a pessimistic picture, in that it does not consider the fact that only about 8 days are really usable because of the lighting constraint. A more realistic picture of the area available for a month, would be obtained if the latitude limits were defined for the longitudinal regions on the days when the lighting was acceptable in those regions. That is, select only those days when the sun elevation is between 7° and 20° for the longitudes between 45° E and 45° W; and furthermore on any one of these days define the latitude limits only for the longitude region which had acceptable lighting. The area available during the month of February 1968 under these conditions is illustrated in figure 29. The available

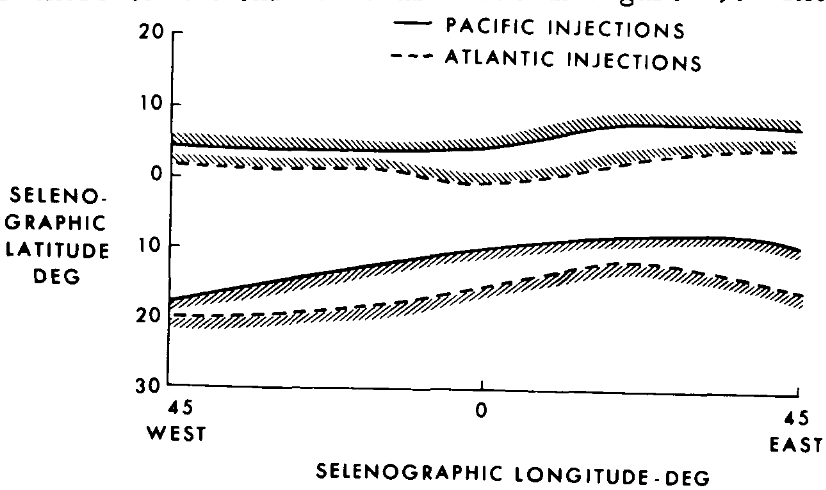

Figure 29.- Accessible lunar landing area in Feb. 1968 when the lighting conditions are correct.

area defined in figure 29 cannot be extrapolated from month to month because the lighting cycle and the declination and libration cycles do not have the same period. Therefore, one of these figures must be made for every month of interest.

The accessible areas for each month during a year can be combined to define an area available for the entire year. This is illustrated in figure 30.

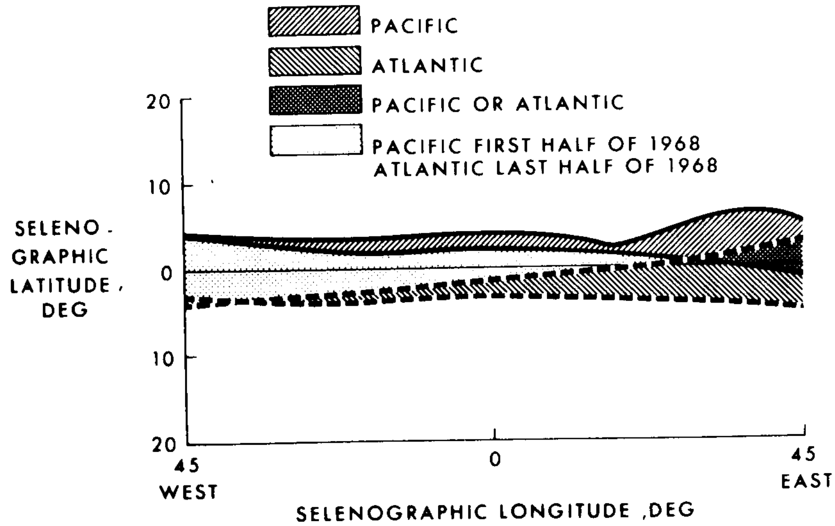

Figure 30.- 100% lunar surface accessibility for 1968.

The purpose of all of this discussion of performance limitations has been to show that the landing site location has a major effect on launch opportunities; not only through lighting conditions and longitude interactions but also through latitude and performance interactions. This total interaction can be summarized as follows: given a landing site location, a launch is possible only on the day that the lighting is acceptable and then only if the landing site is within the latitude bounds attainable for that longitude on that day.

If lunar landing sites could be selected entirely on the basis of performance and lighting constraints, they could be located so that there would be no restrictions on launch windows. Unfortunately, there are many factors that must be considered in the selection of lunar landing sites. These other factors force compromises to be made, with the result being that launch windows are in fact constrained by the available lunar landing sites.

Daylight Launches and Minimum Number of Launch Opportunities During a Month

We have seen that lunar landing sites, lighting requirements, and spacecraft performance are all very effective in constraining the launch opportunities. Just how constraining will be illustrated shortly. In order to illustrate the limitations on launch opportunities, it will be necessary to assume certain lunar landing sites. For the purposes of this illustration, we will assume that the sites to be photographed on the lunar Orbiter A and B missions are found to be acceptable for Apollo landings.

First, let's consider the seven sites to be photographed by Orbiter A. The launch opportunities provided by these seven sites throughout 1968 are summarized in figure 31. The interesting features of this figure are the pattern and frequency of launch opportunities and the frequency of night launches.

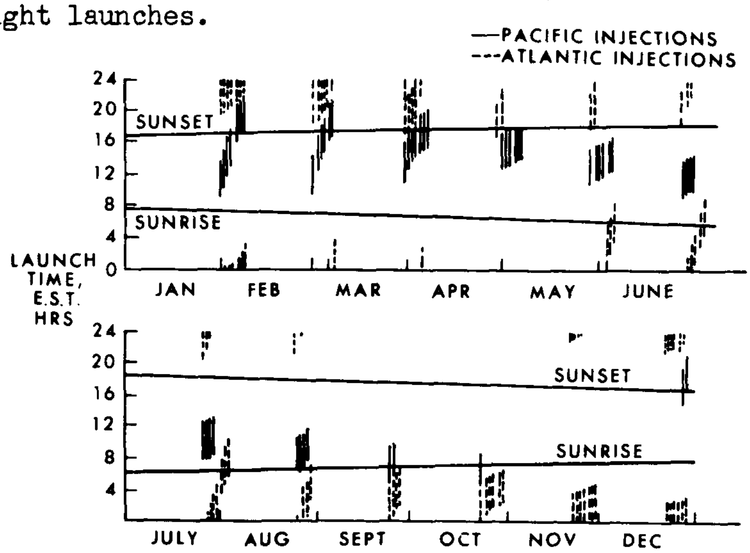

Figure 31.- Apollo launch opportunities in 1968.

During some of the months, many launch opportunities are not shown for the Pacific injection window. This is because some or all of the sites were outside of the performance boundaries and were not available when the lighting was acceptable. The Pacific injections are particularly affected by this performance limitation because the western sights to be photographed by Orbiter A are generally south of the equator, away from the best performance region of the Pacific windows. In the early part of the year, the moon's declination and lighting are in a favorable phase, and the southern sites are well within the performance capability of the Pacific injection. However, in the latter part of the year, the moon is at a northern declination when the lighting is acceptable and the southern sites are outside of the accessible area.

Since the best performance region for the Atlantic injection window is south of that for the Pacific, the lighting and libration combination is favorable for this window in the latter part of the year. The result is that for those months when the Pacific injections are unavailable, the Atlantic injections are available. So launch windows exist all year.

The effect of constraining the mission to be launched only in daylight can also be determined from figure 31. Although this is not considered to be a firm constraint for all lunar missions, it is highly desirable, and every effort will be made to have a daylight launch - at least for the first one.

From figure 31, it can be seen that to limit the launch to daylight hours eliminates the Atlantic injection window for the entire year. In addition, several of the launches using the Pacific injection window occur at night during the winter months and would also be lost.

The net result of the combination of performance limitations and a daylight launch constraint would be virtual elimination of all launch windows in the last quarter of 1968.

The effect of a minimum launch window duration constraint is demonstrated in figure 32. This figure shows the remaining launch opportunities if only those launch windows of 5-days duration were considered.

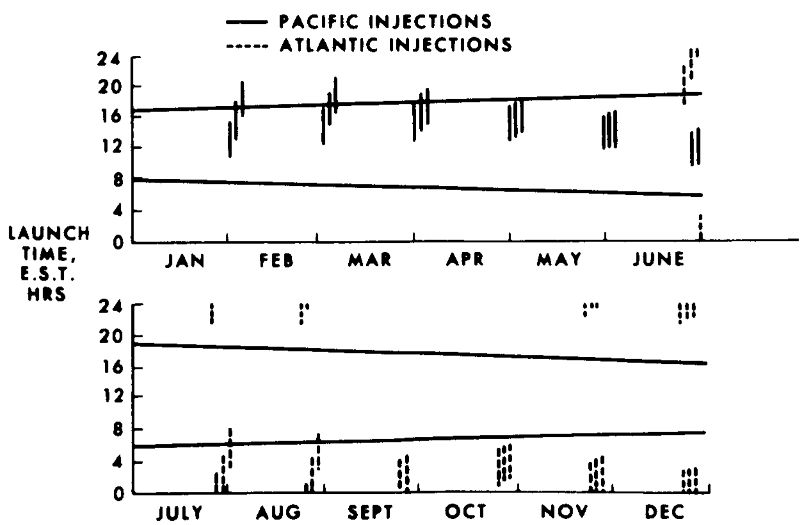

Figure 32.- Apollo launch opportunities in 1968 that provide a 1-3-5 day launch window.

It is clear that the launch opportunities afforded by the seven sites used in this analysis are entirely satisfactory during the first half of 1968. The situation rapidly changes from marginal to unsatisfactory during the third quarter and remains that way for the rest of the year because night launches would be required. Daylight windows of 5-day duration are available in March, April, May, and early June. However, beginning in late June and continuing throughout the rest of year, a night launch would be required.

To improve the daylight launch capability, the sites to be photographed on the Orbiter B mission were located north of the A sites. This provided sites in an area more favorable from the Pacific injection window. The resulting 5-day launch windows are summarized in figure 33. It can be seen from this figure that 5-day windows with a daylight launch could be obtained throughout the year if the Orbiter B sites were available.

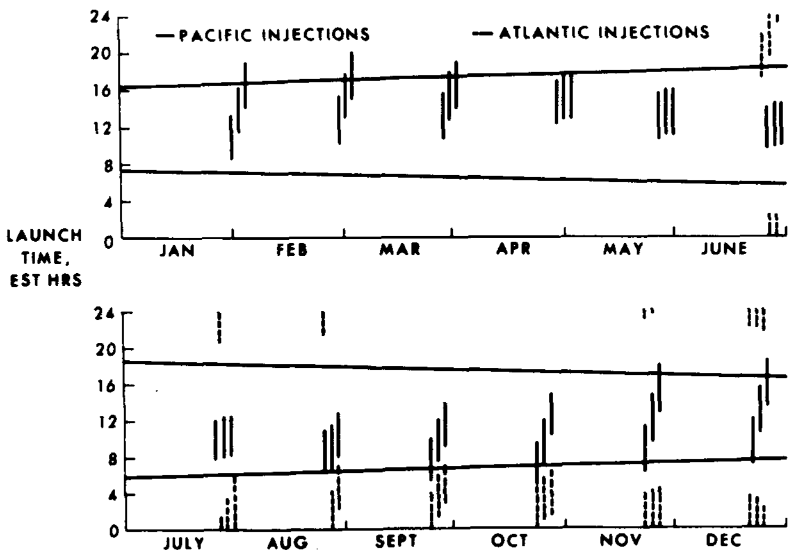

Figure 33.- Apollo launch opportunities in 1968 that provide a 1-3-5 day launch window for Orbiter B sites.

Figures 34 through 40 are similar to figures 24 and 27 through 32, respectively, except that they present information for 1969 instead of 1968.

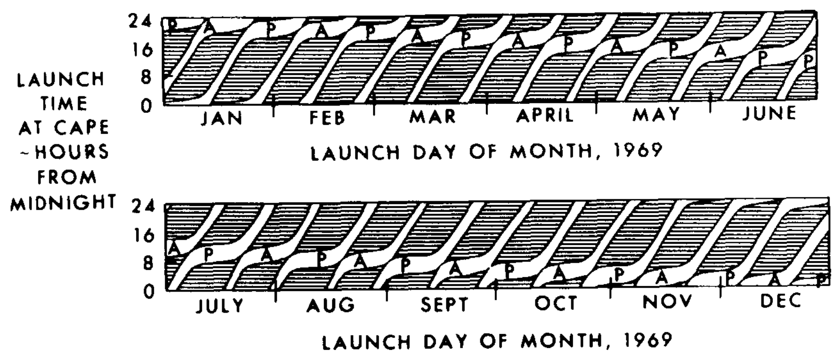

Figure 34.- Effect of launch azimuth constraints in 1969.

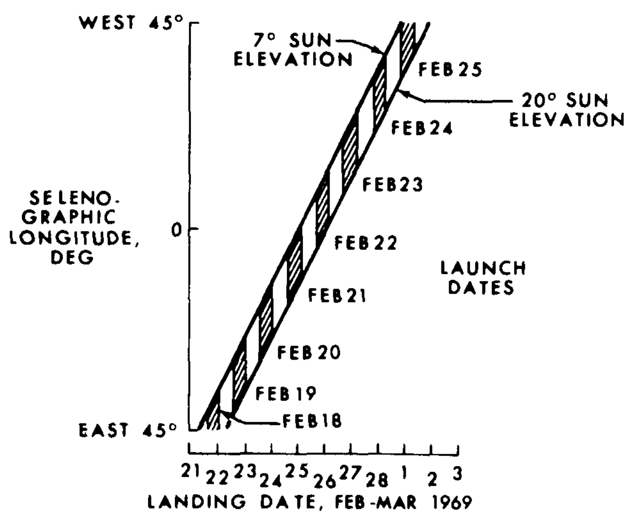

Figure 35.- Lunar landing opportunities.

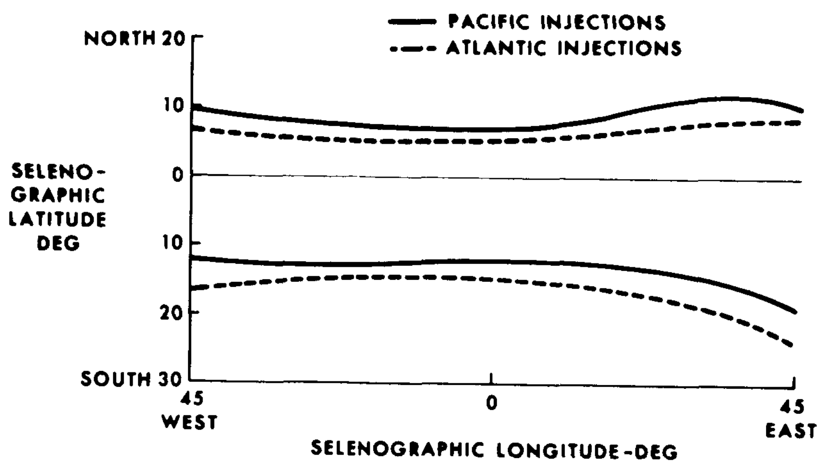

Figure 36.- Accessible lunar landing area for Feb. 18, 1969.

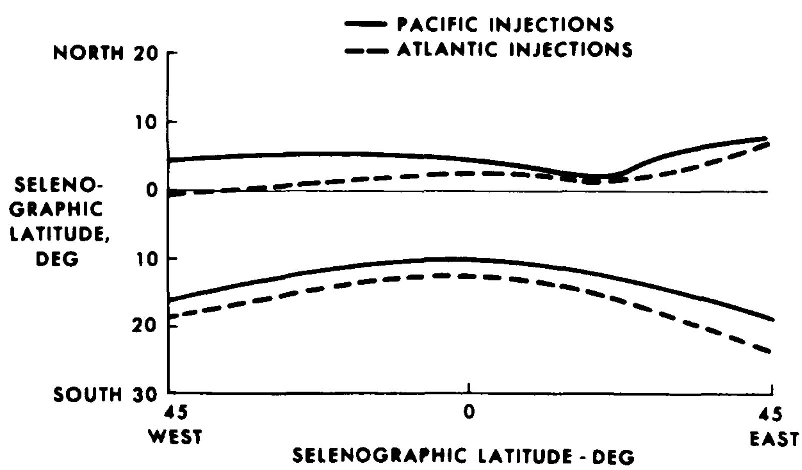

Figure 37.- Accessible lunar landing area in Feb. 1969 when the lighting conditions are correct.

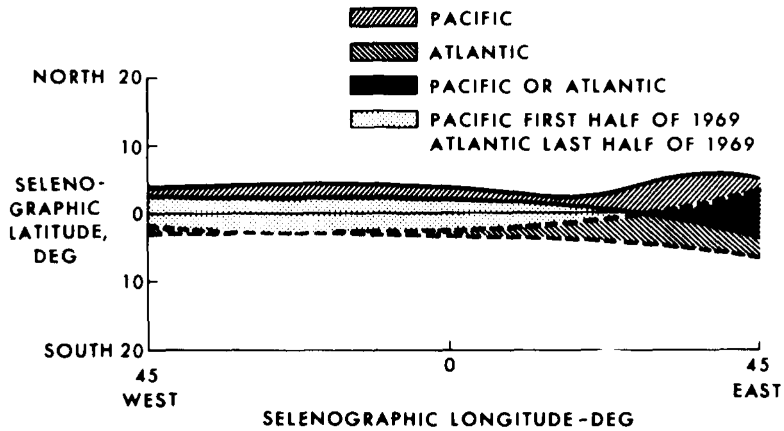

Figure 38.- 100% lunar surface accessibility for 1969.

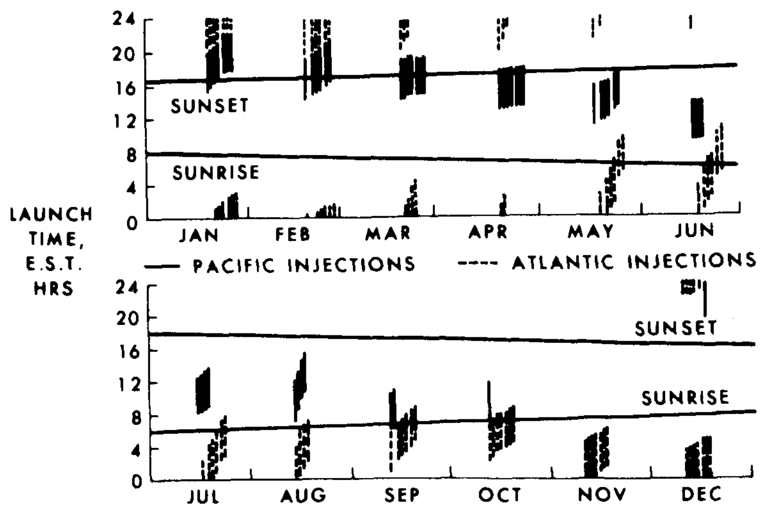

Figure 39.- Apollo launch opportunities in 1969 for Orbiter A sites.

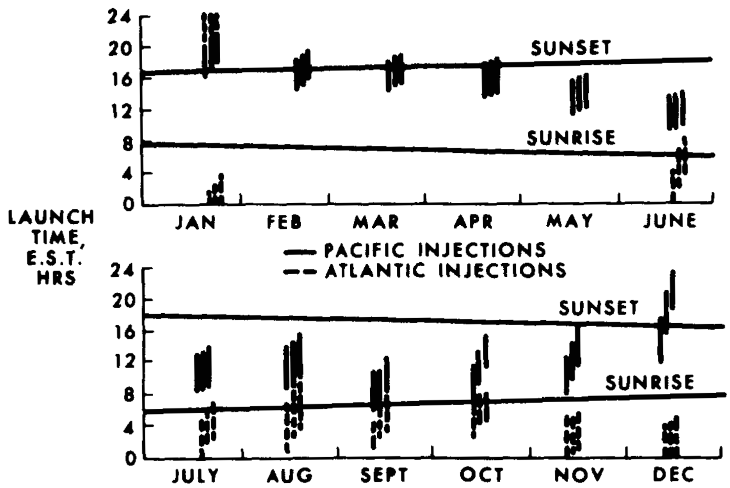

Figure 40.- Apollo launch opportunities in 1969 that provide a 1-3-5 day launch window for Orbiter B sites.

SECOND SESSION: TRANSEARTH INJECTION THROUGH REENTRY

INTRODUCTION

In this section we will finish off the mission planning considerations and the affects of constraints on trajectory shaping of the nominal mission. All we have to discuss on this topic are the problems of getting the spacecraft back from the moon through the reentry corridor to the recovery area. This section is divided into two phases, transearth and reentry.

TRANSEARTH

The geometric restrictions on transearth injection are somewhat similar to those of the translunar injection. However, proper allowances must be made for the fact that the spacecraft trajectory relative to the moon is hyperbolic instead of elliptic and the moon is orbiting about the earth. In the transearth case the target body is relatively stationary and the spacecraft is leaving the body which is in orbit. The spacecraft must break out of the lunar gravitational sphere and fall back to earth. The velocity relative to the moon must be increased in order to escape, but the spacecraft inertial velocity must be decreased in order to return to earth.

Relative velocities are illustrated in figure 41 on the following page. At a distance from the earth equal to the distance to the moon's sphere of influence, the earth relative velocity vector required to obtain a given perigee altitude is illustrated by the vector V_{re}. In this figure two cases are shown; a high energy short-time trajectory and a low energy long transit-time trajectory. There is a continuum of safe return trajectories between the two extremes. The moon's orbital velocity is shown as the vector V_m. The spacecraft must obtain a velocity vector relative to the moon V_{rm} that results in a velocity relative to the earth V_{re}. The minimum velocity relative to the moon occurs when the velocity relative to the earth is a minimum or at apogee of the earth-return trajectory. This is not necessarily the longest flight time because the spacecraft could leave the moon on such a trajectory that has a postive flight-path angle relative to the earth. This type of trajectory is of no interest, however, because the increase of transearth flight time does not allow a decrease in the injection velocity requirements. The conclusions that can be drawn from this figure are that faster return times require larger exit velocities and hence larger transearth injection velocities. It can also be seen that the lower injection velocities require a more nearly retrograde motion upon leaving the moon's sphere of influence.

Figure 42 illustrates the trajectory geometry inside the moon's sphere of influence relative to the exit conditions. In this figure the moon's sphere of influence is approximated by the large diameter circle. The velocity of the moon relative to the earth is given by the vector V_m. Since the sphere of influence moves with the moon any point on this sphere would also have the velocity relative to the earth of V_m. The trajectories inside this sphere are hyperbolic relative to the moon and the energy of these trajectories determines the magnitude of the exit velocity V_{rm}. The direction of this exit velocity is primarily controlled by the exit position which in turn is determined by the longitude at which the injection is performed. To obtain a more retrograde direction of the exit velocity, the injection must occur around on the leading edge of the moon as illustrated by the upper trajectory in figure 42. To obtain a high energy, short time, earth return trajectory the injection maneuver must be performed on the trailing edge of the moon. This

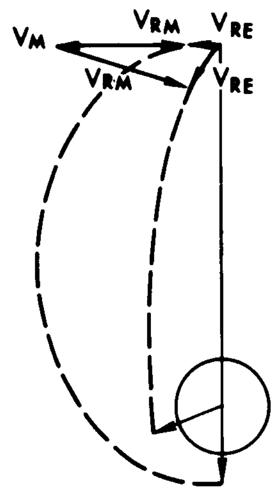

Figure 41.- Effect of lunar sphere exit velocity on return trajectories.

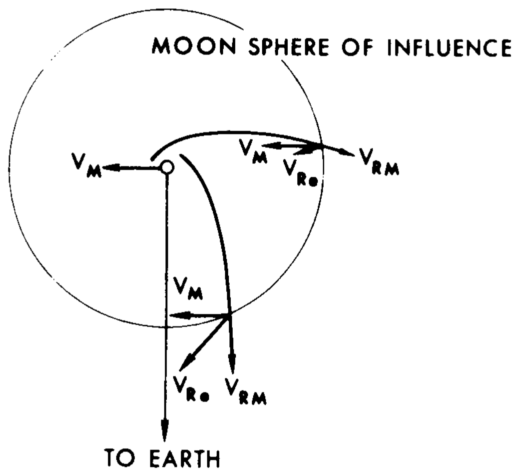

Figure 42.- Lunar sphere exit hyperbolas that return to earth.

results in exit velocities directed more toward the earth.

It can be seen that once a desired transearth flight time has been selected the exit velocity requirement can be determined, and in turn, these exit velocity requirements will specify the location and energy of the transearth injection maneuver. The shaded area in figure 43 shows the region of longitudes where the transearth injection maneuver could be performed. It extends from about 140° W longitude to about 140° E longitude on the far side of the moon.

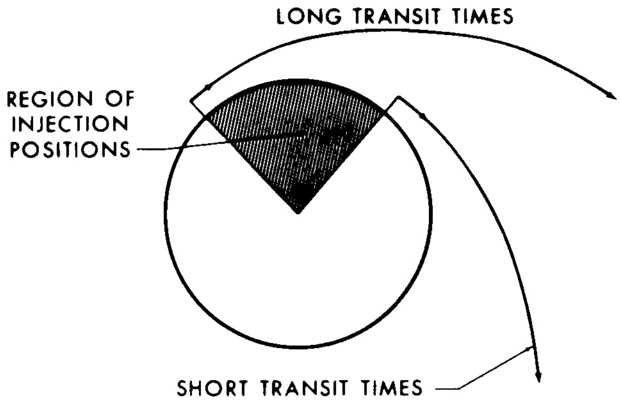

Figure 43.- Effect of transearth flight time on injection position.

The effect of the transearth transit time on the injection velocity requirements can be obtained from figure 44 on the following page, in which the transearth injection ΔV is shown as a function of transit time for two different moon positions in its orbit. The trend of generally decreasing injection velocity requirements with increasing transit time is clearly illustrated. The fact that these curves cross over indicates that there are other factors at work in determining the actual injection velocity required. However, these other factors do not change the basic conclusions.

Now, let's consider how the required transearth flight time is determined. Figure 45 on the following page illustrates the transearth trajectory drawn in the moon orbit plane. The return perigee location is shown relative to the moon's antipode. In this case the moon's antipode is drawn at the time of the transearth injection. This relative angle between the return perigee and the antipode has very little variation with the return transit time. Therefore, the approximate inertial position of the return perigee is a function only of the moon's position. The landing location relative to the return perigee also has very little variation so that in effect the inertial position of the landing is known well in advance of the actual transearth injection.

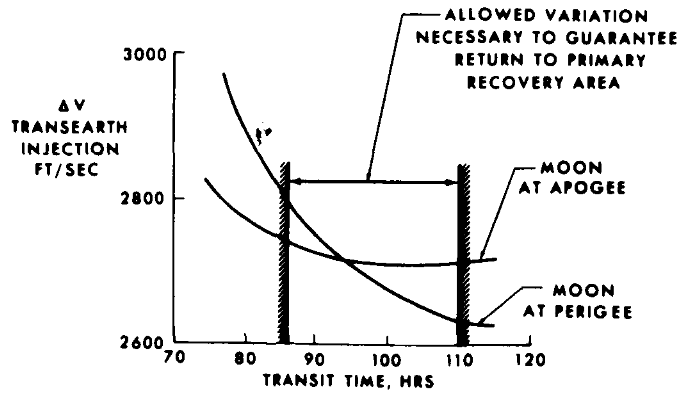

Figure 44.- Effect of transearth flight time on injection velocity requirements.

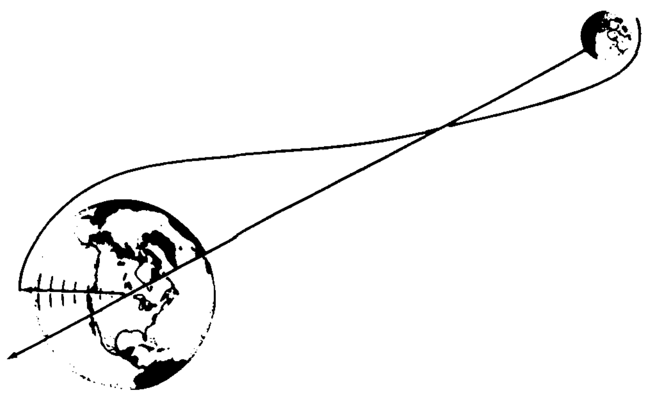

Figure 45.- Transearth trajectory perigee location.

The recovery forces, of course, are fixed to the earth and hence are rotating around with the earth. This leads to the rather interesting situation in which the inertial location of the landing position is well known. However, the inertial position of the recovery forces is highly dependent on the time at which landing occurs. The determination of the transearth transit time is based on the location of the recovery forces. That is, the reentry and landing must occur when the recovery forces are in the proper position. This occurs once every 24 hours, and within any 24-hour range of return transit times, one time can be found which allows rendezvous with the recovery forces.

ENTRY PHASE

For the lunar mission the entry corridor is defined by the variation in flight-path angle at the entry interface altitude of 400 000 ft. Strictly speaking, this allowable variation is a function of the velocity at entry, but there is such a small variation for a nominal lunar mission that the velocity effect is generally omitted. However, for abort returns to earth, this is not the case, and the entry corridor is defined as a function of entry velocity.

The maximum entry angle is defined by the maximum allowable aerodynamic deceleration. Aerodynamic loads encountered during entry increase rapidly with increasingly negative flight-path angles. The high-g side of the corridor is called the undershoot boundary. The minimum entry angle, called the overshoot boundary, is defined by the CM's capability to prevent an uncontrolled skipout of the atmosphere. In addition to being a function of entry velocity, these corridor boundaries are strongly dependent of the L/D ratio of the entry vehicle.

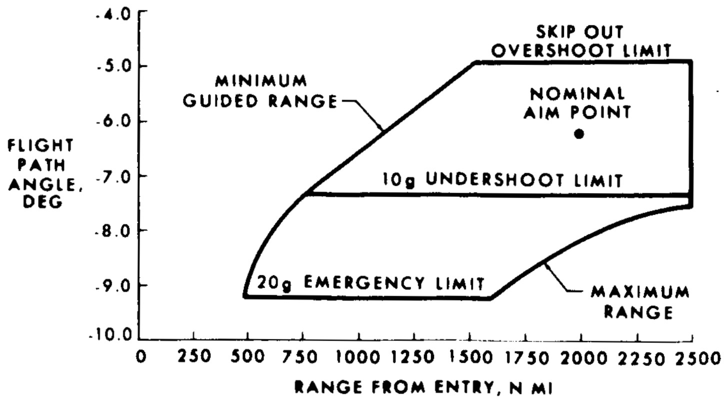

Figure 46.- Entry corridor
Entry speed = 36 000 fps.

Figure 46 shows the entry corridor for the nominal L/D of 0.34 of the CM. In this figure, entry flight-path angle is plotted as a function of range from the entry point to the landing point in order to combine the corridor and maneuvering capability information. The skip-out limit is shown here to be a flight-path angle slightly less than 5°. In other words, any entry conditions at lesser angles than this would result in an uncontrolled skip. The 10-g undershoot limit is shown to be about 7.3°, and any flight-path angle steeper than this would result in loads exceeding 10 g. The emergency limit is defined as a limit below which the aerodynamic load factor would exceed 20 g, which represents the structural design limit of the spacecraft.

The minimum guided range represents the limit to which the reentry range can be controlled with a guidance system. For the overshoot boundary this corresponds to about 1500 n. mi. The maximum range is limited by the lifting capability of the vehicle. At the steeper entry angles the spacecraft could not fly further than this range with maximum positive lift. The maximum range is arbitrarily cut off at 2500 n. mi. to comply with the limitations imposed by the heat shield. The heat shield is designed to tolerate a 3500-n. mi. reentry. There is an entry monitoring system onboard the spacecraft which gives warning when excessive skip is to be encountered. However, the tolerance on this entry monitoring system is about 1000 n. mi. This is made large so that it will not unnecessarily take over a trajectory or give warning that an excessive skip is going to occur. To allow for this 1000-n. mi. tolerance the mission can not be planned with reentry ranges exceeding 2500 n. mi. The inplane maneuvering capability that can be used which would be independent of the entry corridor position is given by the 1500-n. mi. limit and the 2500-n. mi. limit so that ± 500 n. mi. of down-range maneuver capability is available. The nominal aim point represents the conditions that will be targeted to for the transearth trajectory. That is, the return trajectory will be planned to have a flight-path angle at entry of 6.2° and to have the entry point located some 2000 n. mi. away from the landing point. This gives maximum maneuvering capability and allows maximum tolerance of dispersions both in flight-path angle and in range at entry. The primary purpose of the maneuvering capability is to allow a change in the landing site after the transearth injection has been performed. If bad weather were to develop in the area of the recovery forces such that a landing there was undesirable, the spacecraft would have the capability of going 500 n. mi. to either side of this position. Figure 47 on the following page shows the total maneuver footprint, both the downrange and cross-range plotted on a map of the Pacific Ocean hemisphere. The reentry point is some 2000 n. mi. away from the nominal touchdown position. The crossrange capability is about 440 n. mi. at the base and some 660 n. mi. at the toe of this footprint.

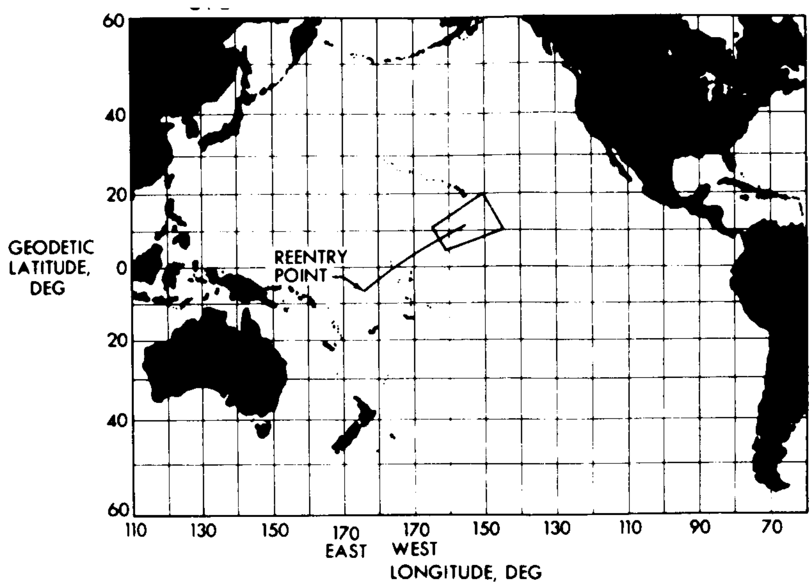

Figure 47.- Operational reentry maneuver capability.

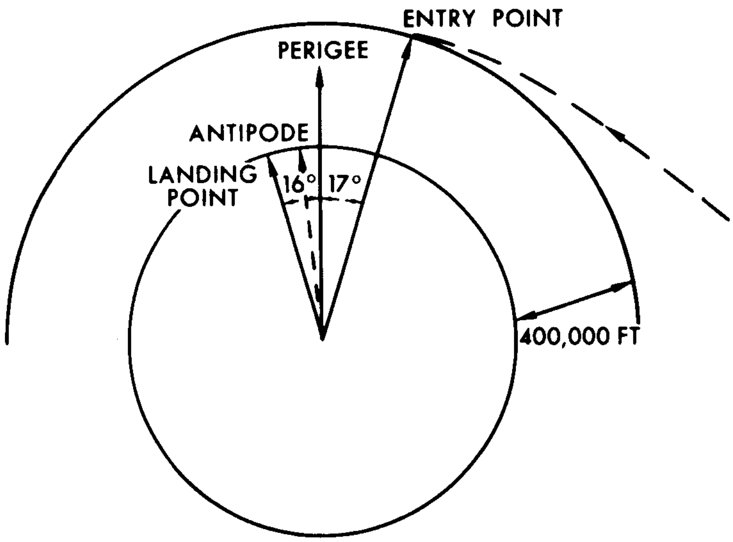

Figure 48.- In-plane trajectory geometry.

Now examine the considerations involved in locating the recovery areas. Figure 48 shows the inplane geometry of the location of significant points of the reentry trajectory. The angular relations shown between the entry point, the perigee of the return trajectory and the landing point have only a slight variation. The location of the return perigee relative to the antipode also has only a slight variation. The result is that the landing point on the earth's surface will be very near the moon's antipode. All of the earth return trajectories will pass through this antipode regardless of the return inclination. Because of the relatively small angle between the antipode and the landing position, very little latitudinal control of the landing can be obtained by varying the inclination of the earth return trajectory. This is illustrated in figure 49. In this figure, two typical earth return trajectories are shown. These two inclinations would be obtained by performing plane changes or performing an azimuth change at the trans-earth injection maneuver. This plan results in a different return inclination for the two trajectories. However, since the antipode is in effect a node through which all of these return trajectories pass and since the landing is fairly close to this node, it can be seen in figure 49 that only a very small amount of latitude variation can be obtained. The amount of latitude control is further restricted by the fact that the return inclination is limited to 40° or less relative to the earth's equator. The landing position control that is available then can be summarized as one in which very fine control is available for the landing latitude. The longitude can be controlled exactly by merely varying the transit time of the transearth trajectory. However,

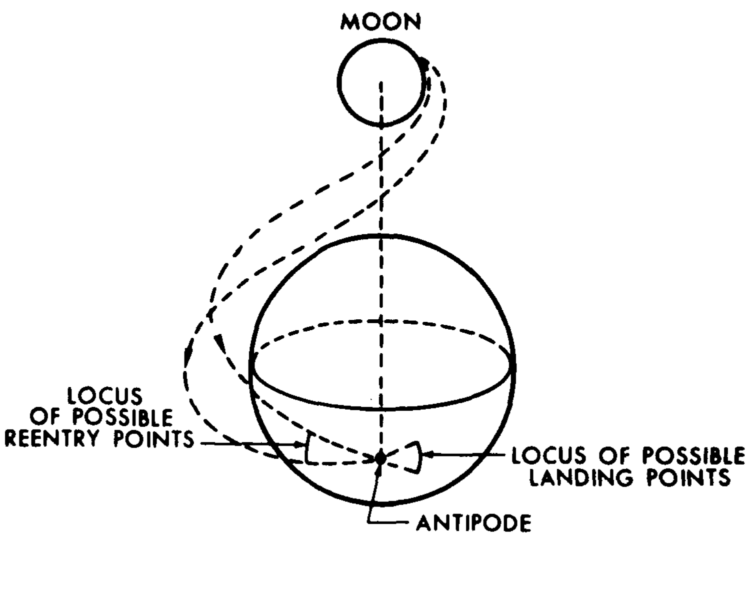

Figure 49.- Effect of plane change on changing the landing point.

the latitude will be a function of the moon's declination at the time of the transearth injection maneuver, and only small variations about this latitude are available by changing the return trajectory inclination. These landing point control characteristics have led to the definition of the recovery zones shown in figure 50.

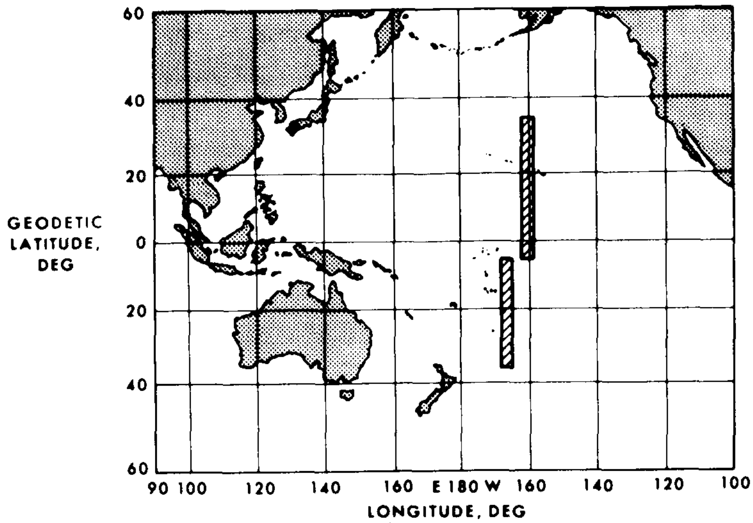

Figure 50.- Nominal recovery areas for mission planning.

There are two zones in which recovery from the lunar landing mission may occur. The northern zone extends from a latitude of 35° N to 35° S along a longitude of approximately 160° W. The southern zone extends from 5° S latitude to 35° S along a longitude of approximately 167° W. The northern zone would be supported by recovery forces staged from Hawaii. The staging base for the southern recovery zone would be Pago Pago in the Samoa Islands. The northern zone extends into the south latitude because of the preference to stage the recovery forces from the Hawaiian base. The shape of these recovery zones reflects the landing area control capability of the return trajectory. The zones need not extend over a large range of longitudes because the longitude can be controlled precisely by the variation in return time. They do extend over a wide range in latitudes since very little latitude control of the landing point is available. The latitude range of these recovery areas is a function of the maximum northern and southern declinations of the moon during any given month.

For any specific mission, of course, it will not be necessary to deploy forces to cover the entire recovery zones. The latitude of the antipode, or the moon's declination, will be confined to a narrow region of latitudes so that the recovery forces for any specific mission will be confined to a narrow region in one of these recovery zones.